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Xenia Mission: Spacecraft Design Concept

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LIST OF ACRONYMS AND SYMBOLS

ACO	Advanced Concepts Office
ACS	Attitude Control System
ARU	array regulation unit
CCAFS	Cape Canaveral Air Force Station
CDS	Command and Data System
CMG	control moment gyro
CRIS	cryogenic imaging spectrometer
DAS 2.0	Debris Assessment Software version 2.0
EDGE	Explorer of Diffuse Emission and Gamma-Ray Burst Explosions
ESA	European Space Agency
FEA	Finite Element Analysis (model)
FEMAP	Finite Element Modeling and Postprocessing (model)
FOV	field of view
GEO	geostationary Earth orbit
GLAST	gamma-ray large area space telescope (now FERMI)
GN&C	guidance, navigation, and control
GRB	gamma-ray burst
GRGT	Guam Remote Ground Terminal
GSFC	Goddard Space Flight Center
HARI	high-angular resolution imager
IMU	inertial measurement unit

LIST OF ACRONYMS AND SYMBOLS (Continued)

K_a	band (26.5–40 GHz)
K_u	Kurtz (band) (11–18 GHz)
Li	lithium
LSP	Launch Services program
MA	multiple-user access
MMH	monomethylhydrazine
NFOV	narrow field of view
N_2H_4	hydrazine
NTO	nitrogen tetroxide oxidizer
RCS	Reaction Control System
SA	spacecraft adapter
SC	spacecraft
STK	Satellite Tool Kit
TDRSS	Tracking and Data Relay Satellite System
TED	transient event detector
T/W	thrust-to-weight (ratio)
WFOV	wide field of view
WHIM	warm-hot intergalactic medium
WSGT	White Sands Ground Terminal

NOMENCLATURE

g	gravity
I	inertia
I_{sp}	specific impulse
M_0	initial mass of the spacecraft
n	number of orbits
z	denotes the Doppler shift for objects receding from Earth
β	angle between the satellite's orbit plane and the Sun
ΔV	change in velocity
γ	target reentry flight path angle

TECHNICAL MEMORANDUM

XENIA MISSION: SPACECRAFT DESIGN CONCEPT

1. INTRODUCTION

Xenia (Greek word for "hospitality") is a concept study for a medium sized astrophysical cosmology mission addressing the Cosmic Origins key objective of NASA's Science Plan. The current situation for new high-energy astrophysics missions both in the United States and abroad is that there are a few major future opportunities available open for competition to a very broad spectrum of scientific disciplines. Under these circumstances, large international alliances encompassing major players in the scientific arena have the best chance of winning. Xenia is being proposed by a large international team to the upcoming Astrophysics Decadal Survey. The Xenia multinational team includes teams from the United States (NASA Marshall Space Flight Center (MSFC)/The University of Alabama in Huntsville/Universities Space Research Association, NASA Goddard Space Flight Center, Penn State, and multiple U.S. universities), Italy, the Netherlands, Japan, France, Denmark, UK, and Germany. The concept of Xenia evolved from the Explorer of Diffuse Emission and GRB (gamma ray burst) Explosions (EDGEs), a mission proposed by a multinational collaboration to the European Space Agency (ESA) Cosmic Vision 2015. Xenia incorporates the European and Japanese collaborators into a U.S.-led mission (Principal Investigator: Dr. Chryssa Kouveliotou, MSFC) that builds on the scientific objectives and technological readiness of EDGE.

The fundamental goal of this mission is to understand the formation and evolution of structures on various scales from the early Universe to the present time (stars, galaxies, and the cosmic web). Xenia will use x-ray monitoring and wide-field x-ray imaging and high-resolution spectroscopy to collect essential information from three major tracers of these cosmic structures: the warm-hot intergalactic medium (WHIM), galaxy clusters, and GRBs. Our goal is to trace the chemo-dynamical history of the ubiquitous warm-hot diffuse baryon component in the universe residing in cosmic filaments and clusters of galaxies up to its formation epoch (at $z = 0–2$) and to map star formation and galaxy metal enrichment into the reionization era beyond $z \approx 6$.

Accomplishing a survey and characterization of the components of the baryonic universe will require high-resolution, soft x-ray spectroscopy and imaging over a wide field of view (WFOV), with extreme low background and the ability to rapidly point at bright GRB afterglows. Currently, no other mission under study will address these issues directly. High-resolution spectroscopy of bright, distant continuum sources in the soft x-ray band will reveal the metals in the WHIM. The expected characteristic emission line intensity requires a high-resolution imaging spectrometer. Bright GRB afterglows will be the beacons. As GRB afterglows fade quickly, rapid localization and repointing capabilities are needed. A wide-field monitor is also needed to carry out follow-up observations with high-resolution spectroscopy. The fraction of the cosmic x-ray background which is due to point sources (active galactic nuclei, or AGN) can be reduced by a factor of ≈3 if these point sources are known or if these can be identified with a high-resolution imaging camera, and the relevant pixels of the imaging spectrometer can be rejected.

The fast repointing (1 deg/sec) requirements of Xenia demand a compact satellite. With a densely packed payload, this is compatible with the Falcon 9 launcher. In order to minimize the effect of the South Atlantic Anomaly on the instruments, mission designers selected a circular orbit 600 km from the Earth at 5 deg inclination.

The mission's lifetime is planned for 5 yr with a possible extension to 10 yr, appropriate to realize the major goals of the mission with a modest guest observer program. MSFC's Advanced Concepts Office, together with an international team of distinguished scientists, performed the Xenia mission concept presented below.

2. SCIENCE MISSION SUMMARY

You, me, the air we breathe, the stars, and the other "normal" matter in the universe are all composed of protons, neutrons, and electrons. These are the ingredients that make all elements from hydrogen to iron and beyond. Astronomers call elements heavier than hydrogen and helium "metals."

Metals are essential for star formation and their subsequent evolution, and ultimately for the formation of planets and development of life as it is known. Understanding the history and evolution of metals is an essential part for our understanding the universe. Using x-ray imaging and spectroscopy, Xenia will read the metal diaries of the universe to explore and reconstruct the cosmic history of metals reaching from the first population of stars to the processes involved in the formation of galaxies and clusters of galaxies.

Xenia's wide-field monitors will watch the sky for GRBs, and when they spot one of these spectacular explosions, the spacecraft will turn and point its telescopes at it in less than a minute. Each burst's brilliance will illuminate the intervening cosmic structures—galaxies, galaxy clusters, and the areas between clusters, together called the cosmic web—for Xenia's wide-field x-ray imager to capture. Then the spacecraft's wide-field spectrometer will identify the chemical fingerprints, or line features, from key elements that help us trace the cosmic chemical evolution. Most "normal" matter in the universe resides in the seemingly empty areas between the galaxies and between the galaxy clusters throughout the cosmic web and is predicted to trace the vast filamentary structures created by dark matter and dark energy. With Xenia's x-ray spectroscopy, astronomers can probe all the metals (carbon through iron) simultaneously, in all ionization stages and all binding states (atomic, molecular, and solid), and thus achieve a unique, model-independent perspective. Xenia will map the gases and collect essential information on galaxy clusters, such as density, temperature, and composition, helping astronomers address the questions listed below.

Xenia will observe and survey, through x-ray telescopes with WFOV and low background (high angular and high spectral resolution), extended sources, like galaxy clusters and the WHIM. Xenia will make observations with a fast reaction to GRBs, thus allowing high-resolution spectroscopy.

Planned as a 5-yr mission, Xenia will address the following fundamental questions:

- When were the first metals created?
- How does metallicity change on cosmic time scales?
- How is the matter in clusters and cosmic filaments distributed?
- What are the physical conditions in large-scale structures?

2.1 Warm-Hot Intergalactic Medium

Due to the unique observational capabilities, Xenia will be able to study the gaseous matter in the universe from the early epochs, through GRB explosions, through the period of cluster formation, down to the present.

High-resolution spectroscopy of bright, distant continuum sources in the soft x-ray band will reveal the metals in the WHIM. Xenia will use bright GRB afterglows as 'backlight' sources, or beacons. GRBs are an unlimited 'renewable resource' and occur out to very large distances, back to the time when the universe was only a fraction of its present age.

Complementary to the absorption spectroscopy, Xenia will image the WHIM and the outskirts of clusters in the emission lines of key elements such as carbon, oxygen, neon, and iron.

2.2 Galaxy Clusters

Galaxy clusters still carry the imprints of primordial cosmological fluctuations and, inside these clusters, prodigious amounts of energy are being converted from one form to another.

Xenia will survey the nearby universe for emissions from a large sample of galaxy clusters. The wide-field imaging and spectroscopy detectors of Xenia will measure surface brightness, temperature, and metal abundances, all the way to the outer regions of many clusters.

2.3 Gamma-Ray Bursts

It is well established that most long-duration GRBs are caused by the explosive deaths of massive stars. Due to their enormous brightness, they can be seen throughout the universe. GRBs produce copious amounts of penetrating high-energy photons and can probe the gaseous regions of the universe, which are not accessible in the optical band.

Xenia will gather a sample of $\approx$400 bright GRB x-ray afterglows in 5 yr, measure their redshift, and identify metal lines (chemical fingerprints) associated with matter along their line of sight. Using high-resolution x-ray spectroscopy, the mission will study the history of metals in both the close GRB environments and their host galaxies back to the early epoch of the universe. Xenia will also quickly relay the GRB coordinates to the scientific community to enable multiwavelength follow-up campaigns.

3. MISSION ANALYSIS

The Xenia spacecraft study required many different areas of mission analysis, including launch vehicle and launch site selection, orbital lifetime estimates, and reentry debris assessment. The team investigated the payload capacities of several launch vehicles, including those currently available and those to be operational within the next few years. Analysts used several industry standard tools, including Satellite Took Kit (STK), Copernicus, and Debris Assessment Software (DAS) version 2.0.

3.1 Mission Requirements and Approach

The mission requirements are summarized in table 1. Given that the preferred launch vehicle was initially the Vega, which was the launch vehicle proposed in a previous study of a very similar spacecraft, constraints suggested that not having a propulsion system could save much needed mass and volume. Therefore, the study team adopted the 600-km value as the initial orbit altitude in an effort to make the orbital lifetime as long as possible, hopefully eliminating the need for periodic orbit boosting.

Table 1. Summary of mission requirements.

Category	Requirement	Notes
Orbit type	Circular	
Orbit inclination	≤10 deg	Minimize the impact of the South Atlantic Anomaly on the science instruments
Orbit lifetime	10 yr	Prefer not to need periodic orbit reboosts
Orbit altitude	500 km required; 600 km goal	Must also meet lifetime requirement

The analyses required several tools as summarized in table 2. While all launch vehicle performance analysis was provided by NASA Launch Services program (LSP), the Xenia team performed the in-space and reentry analysis.

Table 2. Tools used for mission analysis.

Category	Tool	Notes
Launch vehicle performance	Provided by NASA LSP (Vega performance is from Vega User's Manual)	Most payload planners guides do not cover launching into these lower inclination orbits
Orbit geometry	Copernicus	Check orbital elements for 10 yr using 4×4 gravity model
Orbit lifetime	STK and DAS 2.0	
Reentry debris analysis	DAS 2.0	Comply with NASA–STD–8719.14, Process for Limiting Orbital Debris

3.2 Launch Vehicle Performance

Initially, due in part to cost considerations and to the results of a previous design study for a very similar spacecraft, the Xenia science team was proposing the ESA's new Vega as the preferred launch vehicle. Launched from Kourou, French Guiana, this four-stage vehicle with three solid rocket stages and a small liquid vernier fourth stage could place an estimated 2,050 kg into a 600-km circular orbit with a 5-deg inclination. However, reentry analysis (described below) showed that Xenia will need a deorbit propulsion system to ensure a controlled reentry, which increased the mass and volume requirements of the spacecraft. In addition, the science team found it necessary to increase the mass estimates for some of the science instruments. The end result was a spacecraft that would probably not fit on the Vega. Therefore, the analysis team expanded the launch vehicle trade space to include both existing launchers and vehicles that will be operational in the near future, but only those vehicles that will be on contract with the LSP. Some of the vehicles listed in table 3 were chosen based on previous studies involving similar missions, or in the case of the Delta II Heavy, used recently on an actual launch—the gamma-ray large area space telescope (GLAST) spacecraft.

Table 3. Summary of launch vehicle performance.

Parameter	Falcon 9		Vega	Atlas V 401	Delta II Heavy (7920H–10)
Launch site	Omelek (RTS)	CCAFS	Kourou	CCAFS	CCAFS
Source	LSP	LSP	Vega User's Manual	LSP	LSP
600 km @ 5 deg	7,000	1,700	2,050	4,395	895
600 km @ 10 deg	Not requested	2,740	2,040	5,815	1,440
600 km @ 15 deg	Not requested	4,175	Not requested	Not requested	Not requested
500 km @ 5 deg	Not requested	Not requested	2,120	4,390	885
500 km @ 10 deg	Not requested	Not requested	2,110	5,820	1,435

Except for the Vega, the LSP provided performance estimates for all launch vehicles listed in table 3. While the Atlas V 401 provides the necessary mass margin to the desired orbit, the vehicle cost is considerable. The Delta II Heavy, used to launch the GLAST spacecraft, does not provide the necessary payload mass to the desired low-inclination orbit. The Falcon 9, with a scheduled test flight in 2009/2010, seems to be the best choice of the vehicles in the trade space. It offers sufficient payload whether launched from Omelek Island at the Reagan Test Site near Kwajalein, or launched from Cape Canaveral Air Force Station (CCAFS). For these reasons, and the relatively low cost, the team chose the Falcon 9 as the baseline launch vehicle.

3.3 Orbital Lifetime

The required orbital lifetime for the Xenia observatory is 10 yr (see table 1), which was evaluated using the orbital lifetime capabilities of the STK and the DAS. The STK inputs are summarized in table 4. The drag area and exposed areas were estimated by several methods, including using the tools provided with DAS 2.0.

Table 4. Summary of STK inputs for orbital lifetime.

Category	Value
Atmosphere	Naval Research Laboratory Mass Spectrometer and Incoherent Scatter Radar Exosphere released in the year 2000
Solar flux sigma value	2
Rotating atmosphere	No
Drag coefficient	2.2
Drag area	24 m^2
Area exposed to Sun	30 m^2
Reflection coefficient	1
Satellite mass	2,100–2,500 kg
Epoch dates	July 1, 2012 and July 1, 2018
Initial altitude	600 km

The satellite masses used to estimate the orbital lifetime were based on a previous study (EDGE) involving a very similar spacecraft and mission. While the Xenia study resulted in a spacecraft mass of over 2,600 kg, the lifetime results are still valid since this makes the estimated lifetimes more conservative. All things being equal, a heavier spacecraft should have a longer lifetime for the selected orbit.

Figure 1 shows a plot of the orbital altitude as a function of time for the worst of the two chosen epoch dates. Clearly, the orbital lifetime constraints are met. Neither of the two chosen masses results in a reentry before the 10-yr lifetime requirement, and neither has decayed to less than the minimum 500-km altitude in <5 yr. Preliminary results using STK and Copernicus also showed that the orbital inclination and eccentricity do not vary significantly during the 10-yr lifetime.

3.4 End-of-Life Disposal and Debris Assessment

The end-of-life disposal analysis included determining if a deorbit propulsion system is required, and if so, the ΔV required to ensure atmospheric reentry. An assortment of simple two-body calculations using Hohmann transfers, and detailed propagations in Copernicus, were used to determine the required ΔV to allow for a controlled reentry. Table 5 lists the major parameters used in the deorbit analysis.

Given these values, analysts used simple Hohmann transfer calculations to determine the required impulsive ΔV values necessary to target various flight path angles during atmospheric reentry; results are plotted in figure 2.

To estimate the effect of gravity losses for various thrust-to-weight (T/W) values, team members used Copernicus to integrate the equations of motion during a finite burn. The total deorbit ΔV requirement, including gravity losses, is 163 m/s. This value includes an impulsive ΔV to target a Hohmann transfer with a reentry flight path angle of –1.75 deg. The perigee value corresponding to that flight path angle was then put into Copernicus, where the finite burn ΔV was calculated by

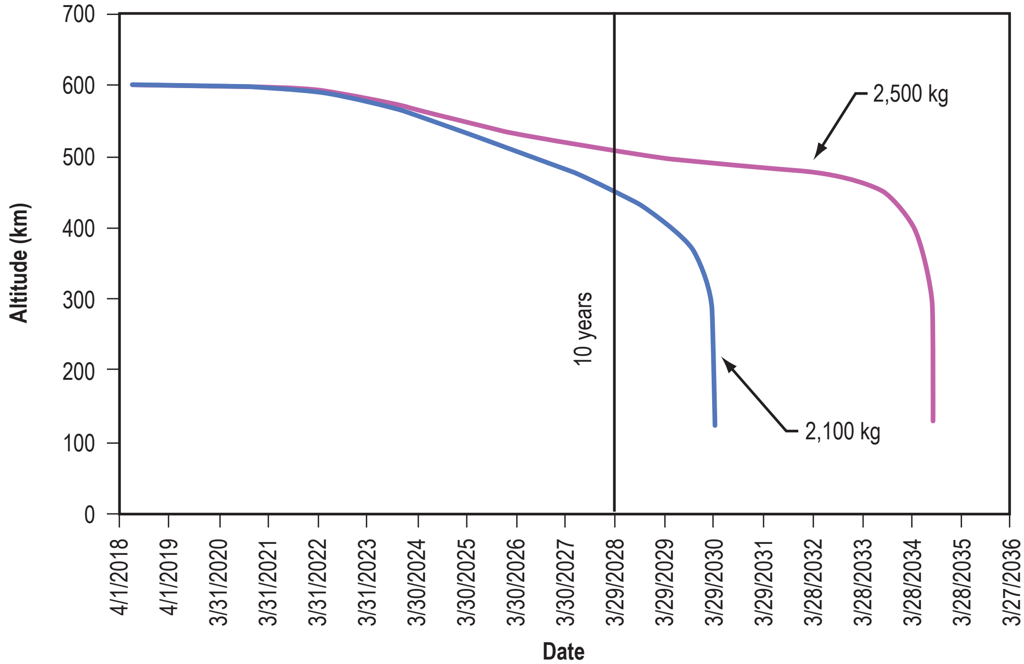


Figure 1. Plot of orbital altitude versus time.

Table 5. Parameters for deorbit analysis.

Category	Value
Orbit shape prior to deorbit	Circular
Altitude of deorbit burn	Worst case between 500 and 600 km
T/W limit for gravity loss	0.025
Reentry altitude	400,000 ft or 122 km (used in Hubble Space telescope end-of-life study)
Target reentry flight path angle, γ	–1.75 deg
Range of acceptable γ	–1.4 to –2 deg
I_{sp} used for finite burns	250 s (conservative value)

targeting that perigee. This yielded the gravity loss. The reason for keeping the gravity loss separate from the ΔV calculation was to allow analysts to clearly see where the gravity losses begin to get large, and thus place a lower limit on the acceptable T/W ratio for the spacecraft. Given the gravity loss values plotted in figure 3, the team set a lower limit on the acceptable T/W ratio of 0.025. This fairly low value allows for a wide trade space for the selection of the propulsion system.

Table 6 summarizes the results of the deorbit analysis. Given the worst-case gravity losses, the conservative reentry flight path angle of –1.75 deg, and a conservative I_{sp} of 250 s, the total ΔV required for deorbit from the 600-km circular orbit is only 163 m/s.

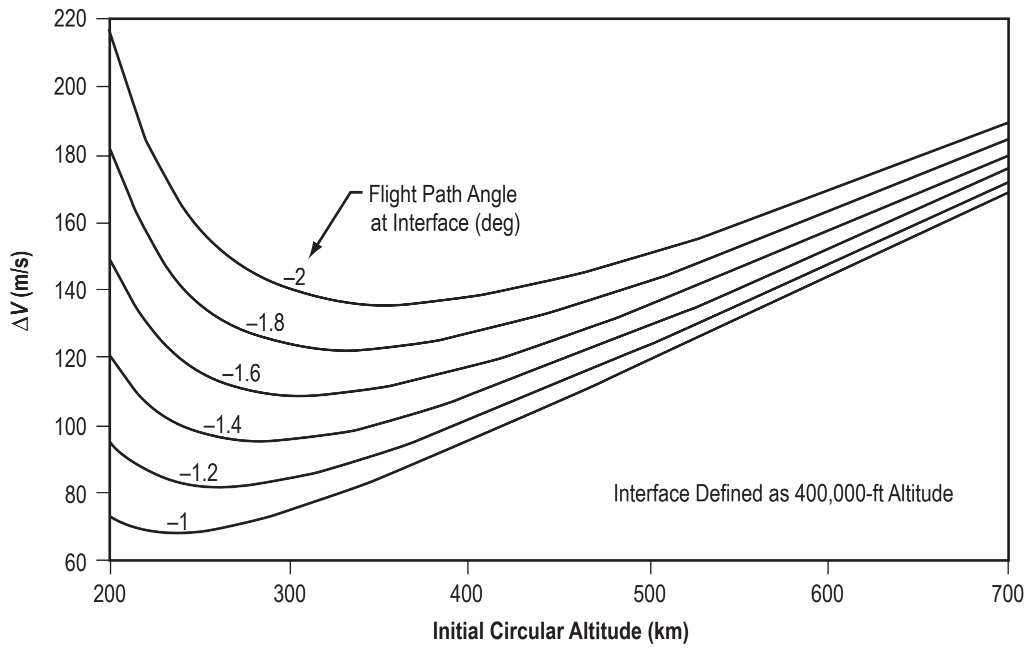


Figure 2. Required deorbit ΔV for various circular orbit altitudes and desired reentry flight path angles.

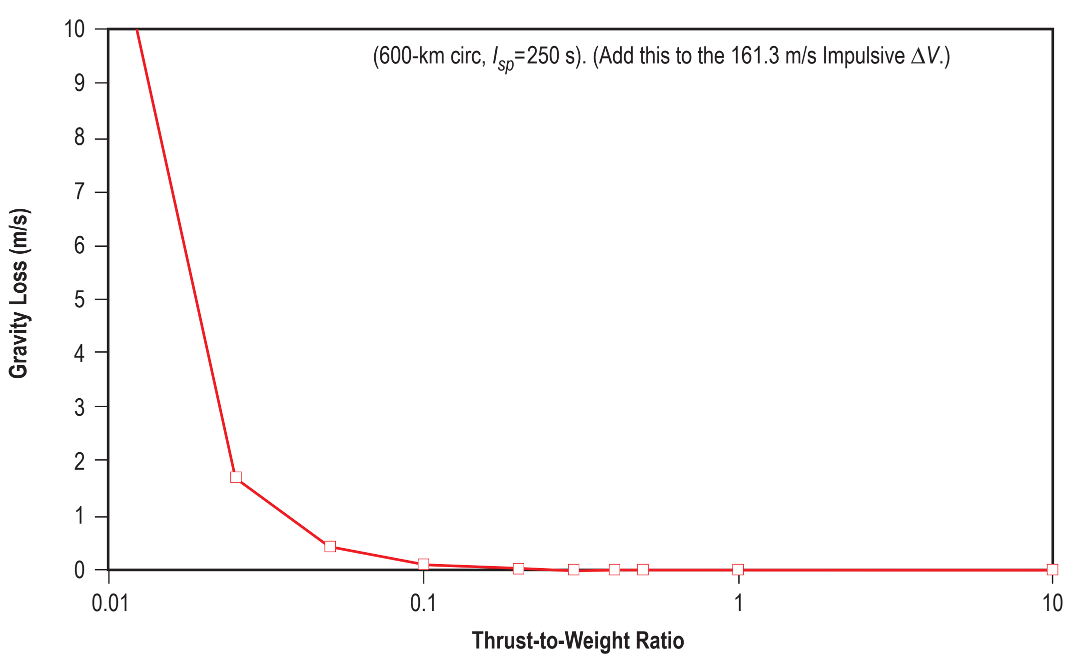


Figure 3. Estimated gravity losses as a function of the T/W ratio.

Table 6. Results of deorbit analysis.

Category	Value
Altitude of circular orbit	600 km
Target reentry flight path angle	–1.75 deg
Impulsive ΔV	161.3 m/s
Gravity loss for T/W of 0.025	1.7 m/s
Total deorbit ΔV	163 m/s

4. CONFIGURATION

The design of the vehicle was driven by the primary science instruments' shape and size. The two large telescopes, the high-angular resolution imager (HARI) and the cryogenic imaging spectrometer (CRIS), were placed each in two parallel axes centered about the spacecraft axis to minimize space and maintain symmetry as much as possible. Both of these instruments were similar in length, which was the driving factor in the overall length of the spacecraft. Their internal component placement, such as the telescope mirrors, were in different vertical locations, so this drove the spacecraft bus structure needed to anchor secondary structural support. With the HARI and CRIS axes centered about the spacecraft axis, the transient event detector (TED) axis could be located in the other two quadrants. This arrangement allowed the TEDs a maximum field of view (FOV) without being obscured by the large scope forward baffles.

As mentioned above, the location of the scope mirrors drove the overall height of the spacecraft bus. The top bulkhead's location also provided a good mounting plane for the TEDs. With the primary instruments set, the spacecraft bus's structure was configured to a rectangular box-like structure to encapsulate the instruments. A center primary bulkhead divided the bus down the middle and allowed for intermediate bulkheads to support the HARI and CRIS at different specialized locations. This panel/bulkhead configuration also allowed excellent mounting points for other spacecraft systems and avionics. The deorbit system was mounted to its own bulkheads in a similar fashion. The tanks and engine share the same bulkheads.

The spacecraft's control moment gyros (CMGs) location was another important consideration of the bus. (See sec. 6, Guidance, Navigation, and Control (GN&C), for detailed information regarding CMG technology.) The operating volume of the CMGs required the main bus structure to be wide enough to ensure clearance of the HARI and CRIS but within the desired spacecraft volume. The CMGs were also located as close as possible to the vertical center of gravity to minimize power and force necessary for their slewing function. The placement of the two solar array supports was also near the center of gravity and the overall spacecraft center of pressure for the same reason as the CMGs. With the main bus configured, the individual avionics boxes were located on both the outside and the inside of the bulkhead structure. The inside boxes are for HARI, CRIS, and TED avionics and were located as close as possible to their respective instrument. A closeout truss structure encloses the intermediate bulkheads which gives the overall bus structure an octagonal shape. This shape allows the solar arrays to be supported and stowed against the sides of the bus and maintains the symmetry of the spacecraft.

The design driver in the configuration of the spacecraft was to locate the instruments to maximize their observational potential. Once this was done, the supporting structure and systems layout was designed to minimize weight and volume (see figs. 4–6).

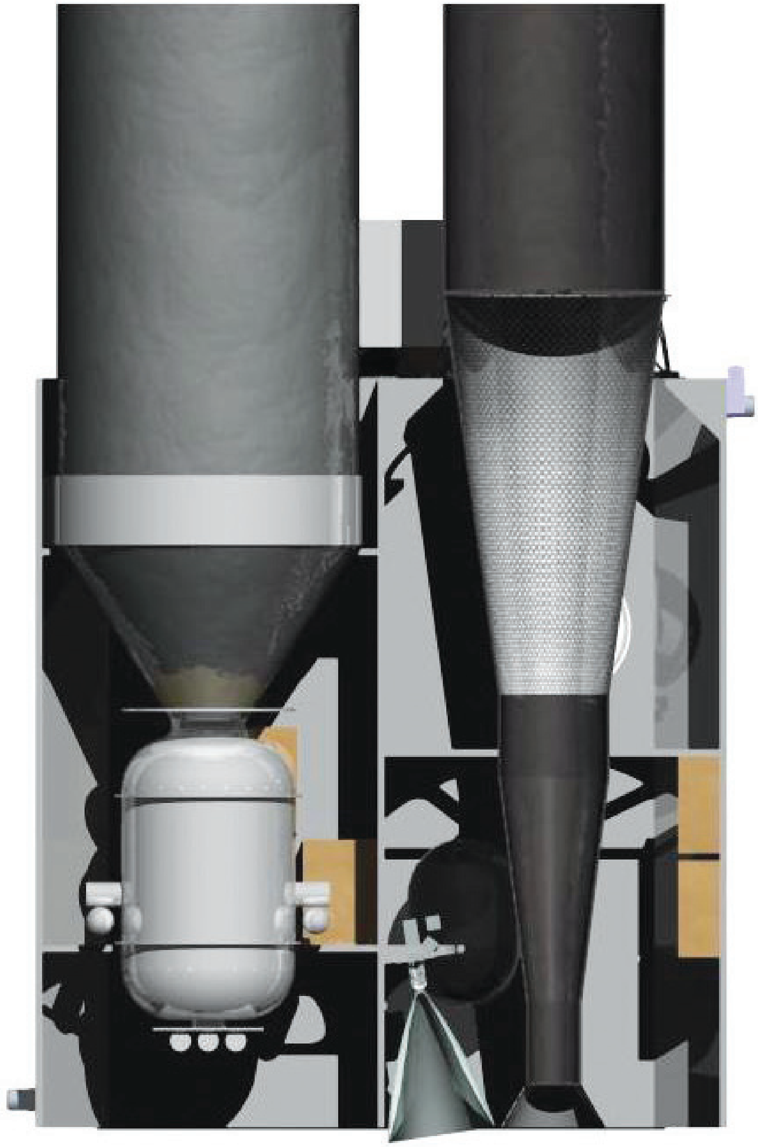


Figure 4. Xenia spacecraft configuration.

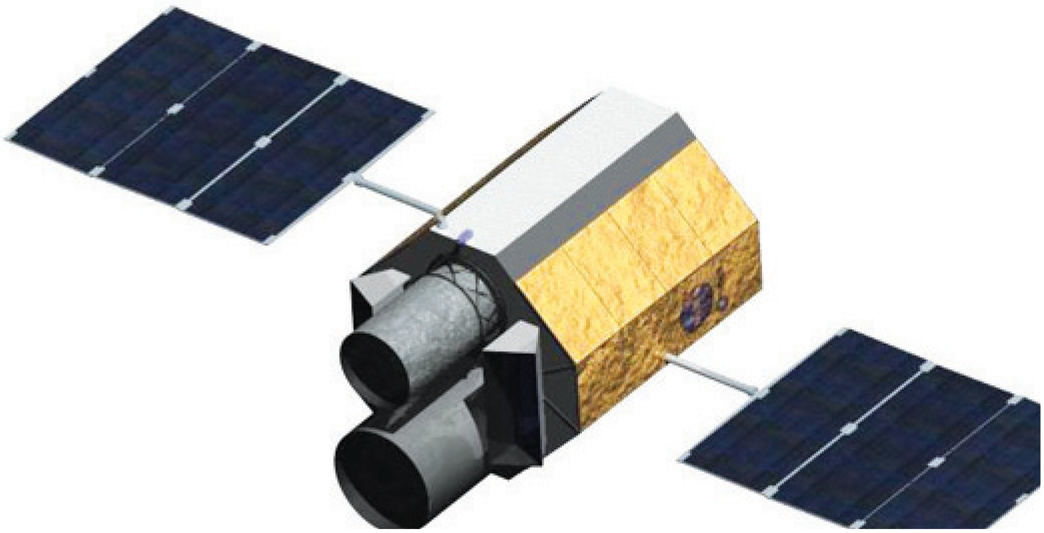


Figure 5. Array.

Figure 6. Shroud stowage concept.

5. MASS PROPERTIES

5.1 Methodology

A collaborative engineering environment with discipline subsystem experts was used to size spacecraft subsystems and propellant loads. Mass properties for the Falcon 9 design used 30% as the accepted growth allowance margin. However, growth allowance for the vehicle design was also calculated without margin and with a growth allowance margin of 20%. Science instrument masses included a 30% growth allowance margin and were provided by the Xenia science team.

5.2 Results

The 30% baseline resulted in a total vehicle gross mass of 2,636.83 kg. Gross mass is the combined total of vehicle dry mass, inert mass, and propellant. Gross mass with no growth allowance margin resulted in a total mass of 2,324.03 kg. A 20% growth allowance margin resulted in 2,535.83 kg of mass.

Dry mass is defined as spacecraft subsystems mass minus the useable propellant, propellant residuals (see 7.0, Noncargo, in table 7), and science instruments. The dry mass total, including the 30% growth allowance margin, resulted in a mass of 1,355.46 kg. Inert mass includes propellant residuals and science instruments. The inert mass totaled 1,144.10 kg. The total less propellant (dry mass plus inert mass) resulted in 2,499.57 kg.

The mass breakdown for the overall vehicle, spacecraft subsystems, and science instruments is shown in table 7.

Table 7. Mass breakdowns.

Weight Breakdown Structure			Quantity	Unit Mass (kg)	Total Mass (kg)
1.0	Structure				399.00
2.0	Propulsion				15.50
3.0	Power				169.52
4.0	Avionics/control				425.94
5.0	Thermal control				32.70
6.0	Growth allowance (30% margin)				312.80
Dry mass					1,355.46
7.0	Noncargo				6.10
8.0	Science instruments				1138
	8.1	CRIS	1	575	575.00
	8.2	HARI	1	384	384.00
	8.3	TED	1	144	144.00
	8.4	Instrument cabling	1	35	35.00
Inert mass					1,144.10
Total less propellant					2,499.57
9.0	Propellant				137.26
Gross mass					2,636.83

6. GUIDANCE, NAVIGATION, AND CONTROL

The Xenia spacecraft will be three-axes stabilized using reaction wheels and CMGs. No propellant Reaction Control System (RCS) will be used. The Xenia spacecraft is to have a 360-deg entire sky operational pointing and viewing capability, with 45-deg Sun avoidance. There is no Earth or Moon viewing avoidance required due to instrument sensitivity. The science team judged that, while an extended Earth view could possibly cause damage to the CRIS detector, slews across the Earth view would be acceptable.

In order to achieve science objectives, a fast slew requirement of 60 deg in 60 s was chosen for the spacecraft. That equates to an acceleration rate of 2 deg/s in 30 s, which is used, along with the vehicle moments of inertia, to drive the torque requirements for the actuators of choice. The science team estimated that a fast slew may be required once every 24 hr.

Initially, a strategy of using two different Altitude Control Systems (ACSs)—one for fast slews and another for slow slews and station keeping—was determined to be a good approach for this mission. Analysis showed that to use reaction wheels to perform the fast slew would require massive wheels and power to achieve the needed torque capability. In addition, torque and momentum storage were inherently conflicting requirements in reaction wheel technology, and this mission calls for the extreme of both ends. This leads one to consider the use of CMGs for the fast slew, as was considered in the EDGE proposal, to meet the high torque requirement and to use an optimized reaction wheel system for the momentum storage requirements of station keeping during long pointing observations.

Although CMG technology for this size satellite is immature or classified, Ball Aerospace has used CMGs (see fig. 7) on their WorldView satellite that launched in 2007; another launch is

- Suggest using Ball Aerospace M95 CMG four-wheel pyramid configuration for all slew, station keeping, and observations.
- Provides up to 6.1 Nm torque (≈4 Nm required for Xenia).

Figure 7. Ball Aerospace M95 CMG four-wheel pyramid set.

planned in 2009/2010. CMGs are suggested for the fast slew requirement of this project, with the acknowledgment that either development or collaboration with Ball or the Department of Defense is needed. The EDGE proposal suggested using six Goodrich reaction wheels 26E2000 with a torque capability of 2 Nm each. This study analysis shows that the required torque for the Xenia fast slews are near 4 Nm for 60 deg in 60 s. Besides providing the required torque, the CMG system has the advantage of requiring less power than a reaction wheel system. The reaction wheel system in the EDGE proposal suggested up to 2.5 kW of required power; CMGs require approximately one-tenth of that power. This power saving alone can justify the use of CMGs for the fast slews. A spacecraft power system to accommodate the 2.5 kW may nearly double in size. The EDGE proposal team opted not to use CMGs due to the technology's immaturity; however, they were unaware of Ball Aerospace WorldView missions at that time.

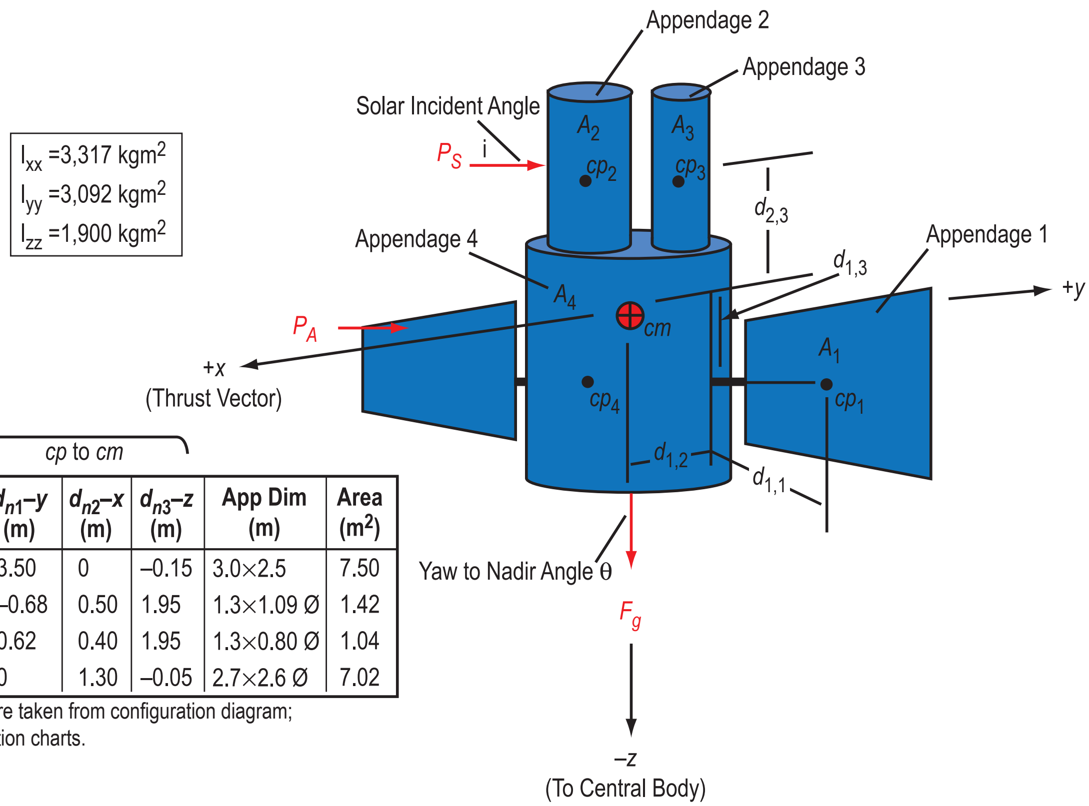
In addition to the fast slew, the spacecraft must have the capability to perform up to five slow slews of 100 deg each per orbit (500 deg total). The slow slew requirements are as follows:

• Five slow slews per orbit.
• 100 deg per slew.
• Observation time up to 30 min.

Within a 90-min orbit, there are five equal 18-min pointing times possible, which must include the slew times between each pointing. Allocating half of the 18 min for the slew leaves 9 min for observation, which means that a 100-deg slew in 9 min is required (or ≈12 deg/min). This is a reasonable slew rate for this size spacecraft, allowing for the use of off-the-shelf reaction wheels. With the moments of inertias as shown on the ACS tool inputs diagram (fig. 8), the torque required for the slow slews is <0.07 Nm.

The L3 company MWA-50 momentum wheels can provide up to 0.07-Nm torque and could be used for the slow slew and station keeping requirements of the spacecraft in a combination configuration with CMGs used for fast slews. However, since the CMGs are available, they can be used for the slow slews in addition to the fast slews. If momentum is perfectly conserved during slew maneuvers—no momentum is lost between the starts and stops of the maneuvers, then theoretically any number of slews could be performed by wheels that could support the torque requirements. However, practical limitations are determined by the saturation period due to environmental torques and slew times. With a slew rate capability of 12 deg/min, two 180-deg slews with a 30-min observation time can be done in one 90-min orbit. Many other pointing scenarios can be done given the various slew times and saturation periods of the traded systems (table 8).

After reviewing the capabilities of the Ball Aerospace M95 four wheel pyramid unit used for the WorldView imaging satellites, analysis showed that this product can meet both the fast and slow slew requirements (along with station keeping demands) without using a reaction wheel set as in the combination scheme. With the loss of one of the CMG wheels, the fast slew capability falls out of the 60-s range down to ≈85 s; whereas, with a combination system, one CMG wheel failure still makes the 60-s slew requirement. This reduced performance, being minor, is acceptable in a failed condition. The full performance in a failed condition is not worth the additional cost, mass, and complexity for a combination system.



	cp to *cm*				
App (*n*)	**d_{n1}–*y* (m)**	**d_{n2}–*x* (m)**	**d_{n3}–*z* (m)**	**App Dim (m)**	**Area (m^2)**
1 - SA	3.50	0	–0.15	3.0×2.5	7.50
2 - CRIS	–0.68	0.50	1.95	1.3×1.09 Ø	1.42
3 - HARI	0.62	0.40	1.95	1.3×0.80 Ø	1.04
4 - SC	0	1.30	–0.05	2.7×2.6 Ø	7.02

Dimensions are taken from configuration diagram;
see configuration charts.

Note:
This diagram used to identify the physical parameters of the spacecraft.

Figure 8. ACS tool inputs.

Table 8. Performance trade.

Number of Wheels	**Source and Type (All Pyramid Configurations)**	**Nominal Wheel Condition**			**One Wheel Failure Condition**			**Masses**		**Power**
		Slew Time (min)	**Science Times (hr)**	**Saturation Period (*n* orbits)**	**Slew Time (min)**	**Science Times (hr)**	**Saturation Period (*n* orbits)**	**Total Wheel Mass (kg)**	**Total System Mass (kg)***	**Total System Power (W)**
8	Four Ball Aerospace CMG-M95 and four Teldix RSI 50-220/45	0.53	3.59	2.5	0.79	1.65	1.2	167.2	334.4	820
4	Ball Aerospace CMG-M95	0.9	3.22	3.8	1.4	0.98	1.48	130.8	261.6	220
6	Teldix MWI 30-400/37	3	1.34	1.44	5	0.79	0.85	91.8	183.6	1,800
4	Teldix MWI 30-400/37	5	0.98	1	15	0.69	0.6	61.2	122.4	1,200

*Total system mass includes isolation mounts and electronics.

The science requirements gave an allowable off-target drift range between 0.75 and 1.25 arc-min. Using the tighter 0.75 value, calculations show that the vehicle will suffer worst-case environmental torques on the *y* axis equal to about 4.32×10^{-3} Nms. Again, using the moments of inertias derived, the disturbance time to set the spacecraft 0.75 arcmin off target will be ≈12.9 s. Judging that an 8.3-s correction maneuver is reasonable and proportionate to the disturbance times, in one orbit period there may be up to 254 correction maneuvers required. For correction maneuvers and disturbance compensation, there is an accumulation of momentum equal to ≈44.1 Nm (table 9). The four L3 reaction wheels have a capacity of 68 Nms. This means that one-and-one-half orbits could possibly be supported before desaturation is required using this L3 system for station keeping alone. However, if they are to perform slews, station keeping momentum needs to be kept in reserve. Giving a 12-deg/min slew rate, 23 Nms will be taken off the station keeping capability and shorten the saturation period to one orbit (90 min). Keeping momentum in reserve allows only 1 hr of science time, which can be divided into any number of slews with the restriction of slew time. For 17-min observation times, three 60-deg slews can be performed in the 60 min less a 5-min slew time for each of the three slews. This type of analysis was done for the four different ACSs traded (table 10 and fig. 9).

Desaturation assumptions can be accomplished using magnetic torque rods. Desaturation times can be significant—a full orbit or more. The spacecraft will have magnetic torque rods installed that can be used for desaturation of both the reaction wheels and CMGs. Desaturation can be assisted by planning of desaturation orientations similar to Hubble operations, which may or may not accommodate science observations. Off-the-shelf magnetic torque rods are suggested from Zarm/Microcosm Company (No. MT400-2); on average, they can provide up to 0.014 Nm of torque each. For the 0.014-Nm torque from the rods to dump 68 Nms would take ≈80 min or approximately one complete orbit. An option to double up the rods on each of the three axis end to end to produce 0.028 Nm of torque to reduce this desaturation time by half was considered. Because of an earlier attempt to make the Vega launch capability, this option was reduced to just one torque rod per axis. The mass of three magnetic torque rods is reflected in the mass summary, one for each axis.

The strategy of having three different attitude control mechanisms—the CMGs, reaction wheels, and magnetic torque rods—is suggested for mission flexibility in meeting science objectives. For example, the reaction wheels could be used to desaturate the CMGs or avoid CMG singularities to perform fast slews in a rapid sequence, subsequently desaturating the reaction wheels with the torque rods while maintaining a target acquisition using the CMGs. Although there is one redundant reaction wheel and CMG, the potential loss of two wheels in one system can be offset using the other system. However, because the Ball Aerospace GMCs were judged to be sufficient alone, as stated earlier, only the CMGs and the torque rods were used in this design.[2]

The following list outlines requirements for pointing accuracy, knowledge, and duration:

- Target tracking up to 30 min at a time.
- After fast slew—within 2 arcmin after 20-s maximum solar array damping time.
- After slow slew—within 1.25 arcmin (undefined).
- Pointing knowledge of 2 arcsec, maintained throughout maneuvers.
- Pointing drift between 0.75 and 1.25 arcmin over 30 min.

Table 9. ACS tool outputs and analysis.

Note: Desaturation period = 1 orbit (≈90 min)	Units	3,317 I_{xx} (kgm^2) (roll)	3,092 I_{yy} (kgm^2) (pitch)	1,900 I_{zz} (kgm^2) (yaw)
Disturbance torques				
Solar torques	Nm	8.195×10^{-6}	1.245×10^{-6}	8.958×10^{-7}
Atmospheric torques	Nm	1.707×10^{-3}	1.153×10^{-3}	1.703×10^{-4}
Gravity torques	Nm	2.604×10^{-3}	2.190×10^{-3}	4.134×10^{-4}
Total disturbance torques	Nm	4.319×10^{-3}	3.345×10^{-3}	5.847×10^{-4}
Disturbance times (0.75-arcmin drift)	s	12.94	14.20	26.63
Disturbance times (1.25-arcmin drift)	s	16.71	18.33	34.37
Attitude corrections				
Correction torque (0.75-arcmin drift)	Nm	0.0417	0.0389	0.0239
Correction torque (1.25-arcmin drift)	Nm	0.0250	0.0233	0.0143
Correction time (0.75-arcmin drift)	s	8.333	8.333	8.333
Correction time (1.25-arcmin drift)	s	13.88	13.88	13.88
Correction momentum (both drifts)	Nms	0.174	0.162	0.099
Correction cycles per desaturation period (0.75-arcmin drift)	No.	254	240	155
Correction cycles per desaturation period (1.25-arcmin drift)	No.	177	168	112
Momentum per desaturation period (0.75-arcmin drift)	Nms	44.11	38.86	15.42
Momentum per desaturation period (1.25-arcmin drift)	Nms	30.74	27.20	11.14
Maneuvers per desaturation period (one each):				
Fast slew torque (60 deg/45 s)	Nm	6.861	6.396	3.930
Fast slew torque (60 deg/60 s)	Nm	3.860	3.598	2.211
Fast slew torque (60 deg/90 s)	Nm	1.715	1.599	0.983
Fast slew momentum (60 deg/45 s)	Nms	154.38	143.91	88.43
Fast slew momentum (60 deg/60 s)	Nms	115.79	107.93	66.32
Fast slew momentum (60 deg/180 s)	Nms	77.19	71.95	44.22
Sum of greatest momentums	Nms	198.49	182.76	103.85
Sum of midlevel momentums	Nms	159.90	146.79	81.74
Sum of least momentums	Nms	121.30	110.81	59.64

Recommended design parameters.

Ball Aerospace M95 CMG: 129 Nms each wheel × 2.31 for a four-wheel pyramid gives 298 Nms, collective torque capability = 6.1 Nm.

To achieve and maintain the pointing accuracy requirement of 2 arcmin, the CMG wheels will be employed, along with solar array dampers, as described in section 8, Power. It is estimated that the dampers will achieve the 2-arcmin damping within ≈15 s after a fast slew. The CMG wheels will then maintain a pointing accuracy within 0.75 to 1.25 arcmin during all observation periods, which will most likely be anytime a slew is not being performed.

Table 10. Number of slews possible per desaturation cycle given equal observation times.

Total Observation Time (min)	System Trades			
	A	B	C	D
17	13	11	5	3
12	18	16	7	5
10	22	19	8	6
8	27	24	10	7
5	43	39	16	12

Note: Momentum required for station keeping and slew times are accounted for.

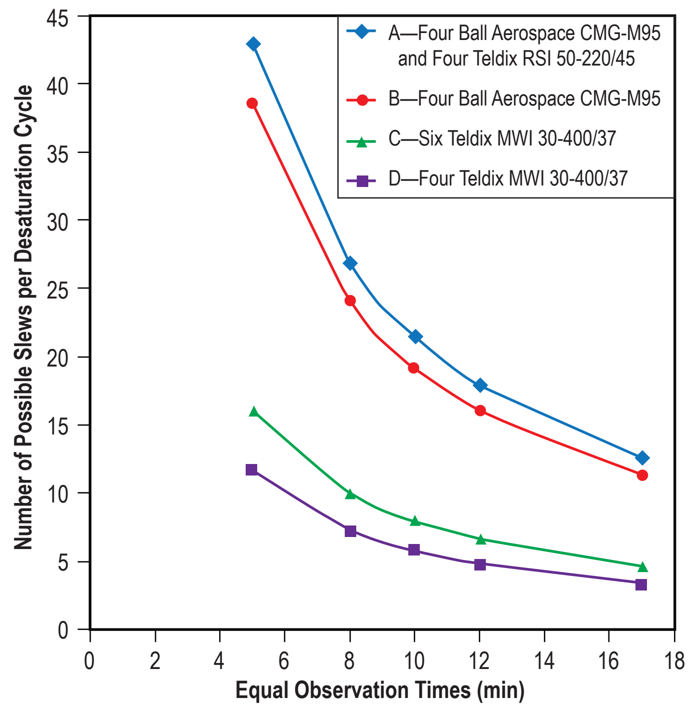


Figure 9. Number of slews possible per desaturation cycle.

To maintain a pointing knowledge of 2 arcsec, the use of two sets of star trackers is suggested. One set of two trackers, narrow field of view (NFOV) star trackers, can be used to get the 2-arcsec pointing knowledge. A Goodrich HD1003 star tracker with an 8-deg FOV was selected for this requirement. However, it is rated to track at a rate of only 0.1 deg/s, and may lose pointing knowledge during the fast slews. To eliminate this possibility, a second set of two WFOV star trackers is used. These are AeroAstro mini-star trackers with a 30-deg FOV; they can track at a rate of up to 10 deg/s. In addition to the star trackers, two inertial measurement units (IMUs) will be on board. The IMUs will work in concert with the star trackers to provide robustness and redundancy in pointing knowledge. The IMU selected is a Honeywell HG9900 unit with a 7.2-arcsec/sqrt hr random walk.

Alternative performance trades were done for sizing the CMGs and reaction wheels, and the following four configurations were considered:

• A combination set of four Ball CMG-M95 and four Teldix RSI 50/220.
• Four ball CMG-M95 alone.
• Six-wheel set of Teldix MWI 30/400 wheels.
• Set of four Teldix MWI 30/400 wheels (see table 8) and science time versus slew time (fig. 10).

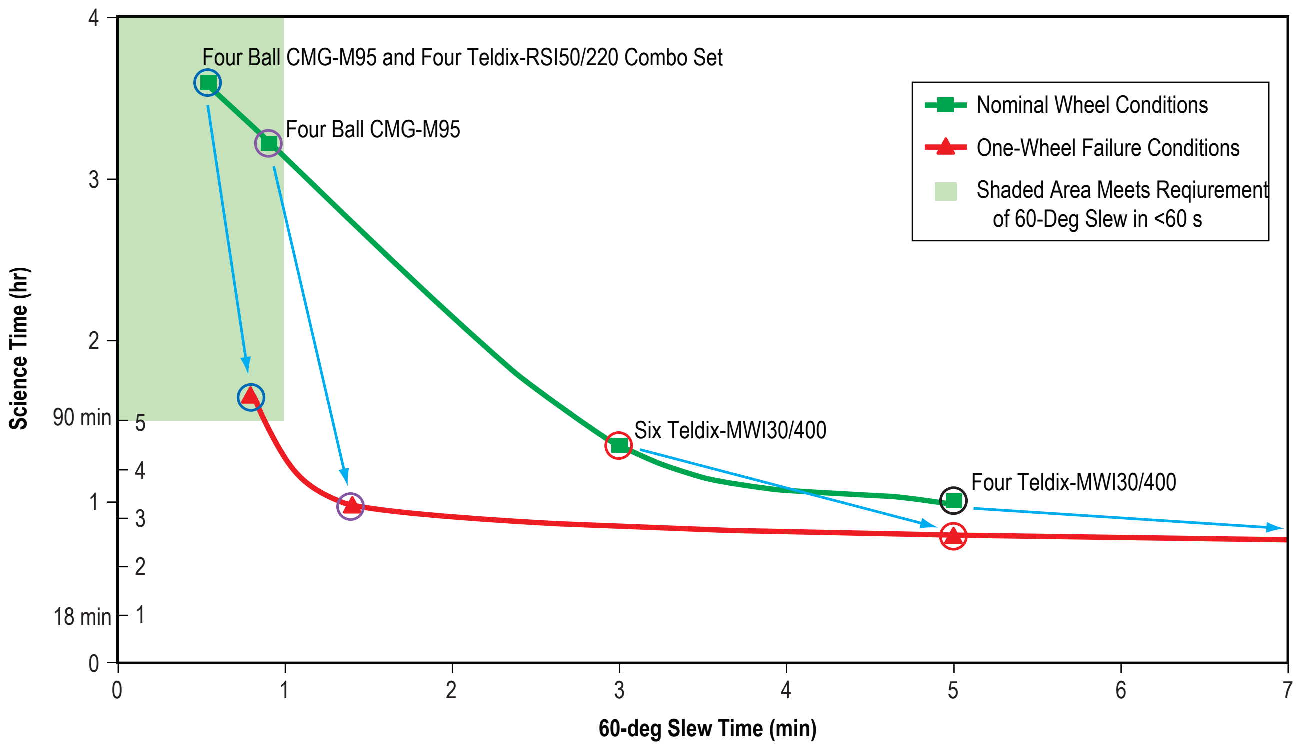


Figure 10. CMG versus reaction wheel performance trade: science time versus slew time (all pyramid configurations).

The trades were done comparing CMGs and reaction wheel sets to perform all the mission functions including the fast and slow slews, along with station keeping, individually and in combination. Using a dedicated set of reaction wheels to perform just station keeping keeps the CMGs in a ready state for target acquisition and allows the fast slew requirement to be met even with the loss of a wheel. However, there is a penalty of greater power, mass, and complexity. The CMG four-wheel set alone meets the 60-s requirement, with a slightly reduced performance with the loss of a wheel. This is a good choice for performing science objectives. The six and four reaction wheel sets are eliminated because they do not meet the 60-deg slew in 60-s requirement, and they have high power requirements.

The fast 60-deg slew maneuver was analyzed at three different performance levels (see table 9). The first level analyzed was at the 60-s slew time, with the settling time considered after 60 s. The next level was analyzed with the 15-s settling time taken off the 60-s slew time (giving a 45-s slew requirement), and the final level analyzed another trade with 30 s added onto the 60-s slew time (giving a 90-s slew requirement). Using the combination set, the four CMGs can almost support the 45-s slew (coming in at 53 s). Since the wheel torques are a function of the inverse time squared, the wheel mass of reaction wheels required would be roughly twice as much for a 15-s faster slew and about half as much for a 30-s slower slew.

6.1 Guidance, Navigation, and Control Results and Conclusions

Using Ball Aerospace WorldView control moment gyro four-wheel set is suggested:

- One set of four CMG wheels to perform the fast slews, slow maneuvers, and station keeping.
- Wheels mounted in a pyramid configuration near the spacecraft center of mass.
- Slightly better performance can be achieved using a CMG and reaction wheel combination set, but this would mean higher mass and power and be significantly more complex. The 60-deg slew in 60 s can be met using a combination set even with a wheel failure.

A set of magnetic torquer rods used to perform the desaturation of the wheels is suggested:

- Desaturation period is approximately one orbit using one Zarm/Microcosm MT400-2 rods, with 0.014-Nm average torque capability per orbit.
- Using two Zarm/Microcosm MT400-2 rods. Doubling the rods end to end, with 2×0.014 Nm $=0.028$-Nm average torque capability per orbit reduces the desaturation time in half.
- Desaturation can be done during science time under certain conditions.

Using two sets of star trackers is suggested:

- One set of two NFOV star trackers, used for the high-accuracy pointing knowledge (2 arcsec). One tracker is used for the x and y axis and a second tracker is used for the z axis. Goodrich has stated that the HD-1003 next-generation star tracker can achieve 1-arcsec accuracy in x and y. In reality, the 1-arcsec knowledge may be hard to achieve in a cost-effective manner (2 arcsec is presently achievable).
- Another set of two WFOV trackers is suggested for maintaining orientating knowledge during fast slews. AeroAstro mini-star tracker has a 10 deg/s rate capability advertised.

7. AVIONICS

7.1 Methodology and Approach

This section summaries the avionics methodology and approach taken. Communications requirements include a total science downlink communication data rate of 3.8 Mbps orbital average and a total telemetry downlink communication rate of 4 kbps per transmission.

The design approach for sending science data to ground is to utilize an omnidirectional broadcast to the Tracking and Data Relay Satellite System (TDRSS). This approach keeps the communication system simple and lightweight. With the spacecraft constantly slewing to new targets, a high gain pointing antenna would be heavily tasked, and vehicle shadowing might occur. Not having a high gain pointing antenna eliminates failure modes and keeps cost down. In addition, the science data will be linked to the TDRSS in the single access mode for periodic transmissions. In this mode, the user must schedule in advance the link times and durations. The link then becomes dedicated to the user, and no interruptions, losses, or delays occur. The standard burst duration in this mode is 10 to 15 min, once per orbit. With a 10-min burst available, the 4-Mbps orbital average data will need to be transmitted at a rate of 36 Mbps as indicated in the data rate versus burst time graph (fig. 11) and table 11 for megabit per second rates and burst times. The TDRSS can support up to 300 Mbps in this mode in the Kurtz (K_u) band (fig. 12).

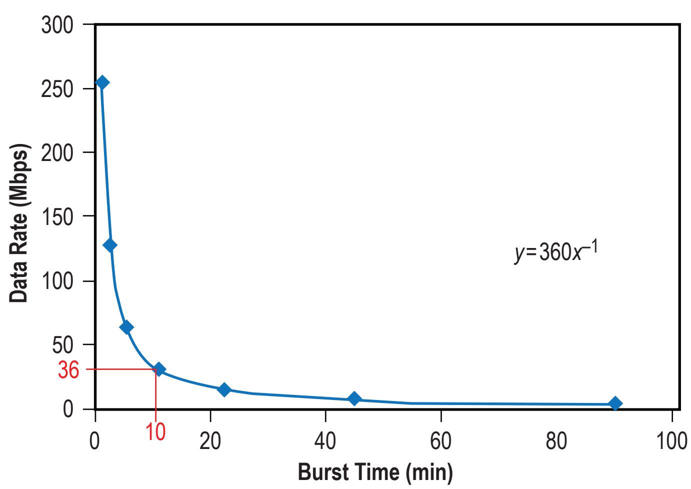


Figure 11. Data rate versus burst time.

Table 11. Megabit per second data rates and burst times.

Rate (Mbps)	Burst Time (min)	Burst Time (s)
4	90	5,400
8	45	2,700
16	22.5	1,350
32	11.25	675
64	5.625	337.5
128	2.81	168.5
256	1.405	84.3

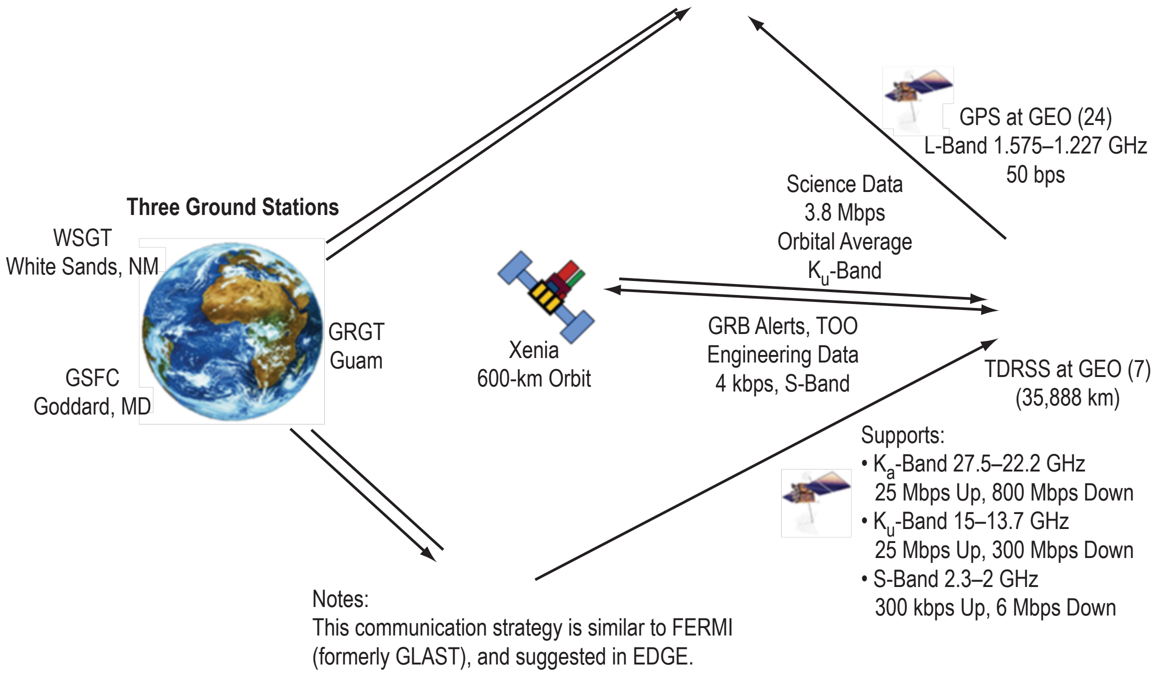


Figure 12. Xenia communications with the TDRSS.

A link budget was performed for omniscience data links to the TDRSS at 10-W transmission power, showing more than sufficient bit error rate margin. The 10-W L3 company T-720 K_u-band transmitter was selected for the primary science communication link. In TDRSS continuous multiple-user access (MA) mode, the burst durations are limited to 2.5 min without significant delays. With this duration, the data rate for a science dump would be 144 Mbps. This is still within the TDRSS capability, but would likely cause scheduling difficulty. Also, the T-720 transmitter selected for the study only supports 75 Mbps maximum. Therefore, the MA mode is not recommended for science data dumps. However, for the other satellite communication requirements—command uplinks, telemetry, and GRB event broadcast downlinks, the MA mode will be utilized in the S-band. This method can achieve a GRB event notification to ground within ≈1 min of the event. Fast, worldwide notification of GRB events is critical for ground operation response. A redundant 5-W AeroAstro S-band radio system was selected for these functions. Both the L3 and AeroAstro radio systems have good in-space heritage and are well suited for these applications.[3]

An avionics requirement for this mission is that 4 Gbit of total science onboard memory is required. To meet the GN&C, command, data management, and communication needs of the spacecraft, a single, multipurpose computer was selected form Saab Ericsson Space division (fig. 13). This computer has a two-bus internal redundancy system for all common satellite flight control requirements. These included ACS boards, propulsion control boards, and instrumentation boards. It also possesses data-processing boards for decoding and formatting commands, telemetry, and science data in preparation for transmission. In addition, it has 2- to 16-Gbit memory boards for a total of 32 Gbit, with spare slots for an additional 32 Gbit of memory. This computer has flight heritage history in several ESA missions, and is considered a very good candidate for the Xenia mission.

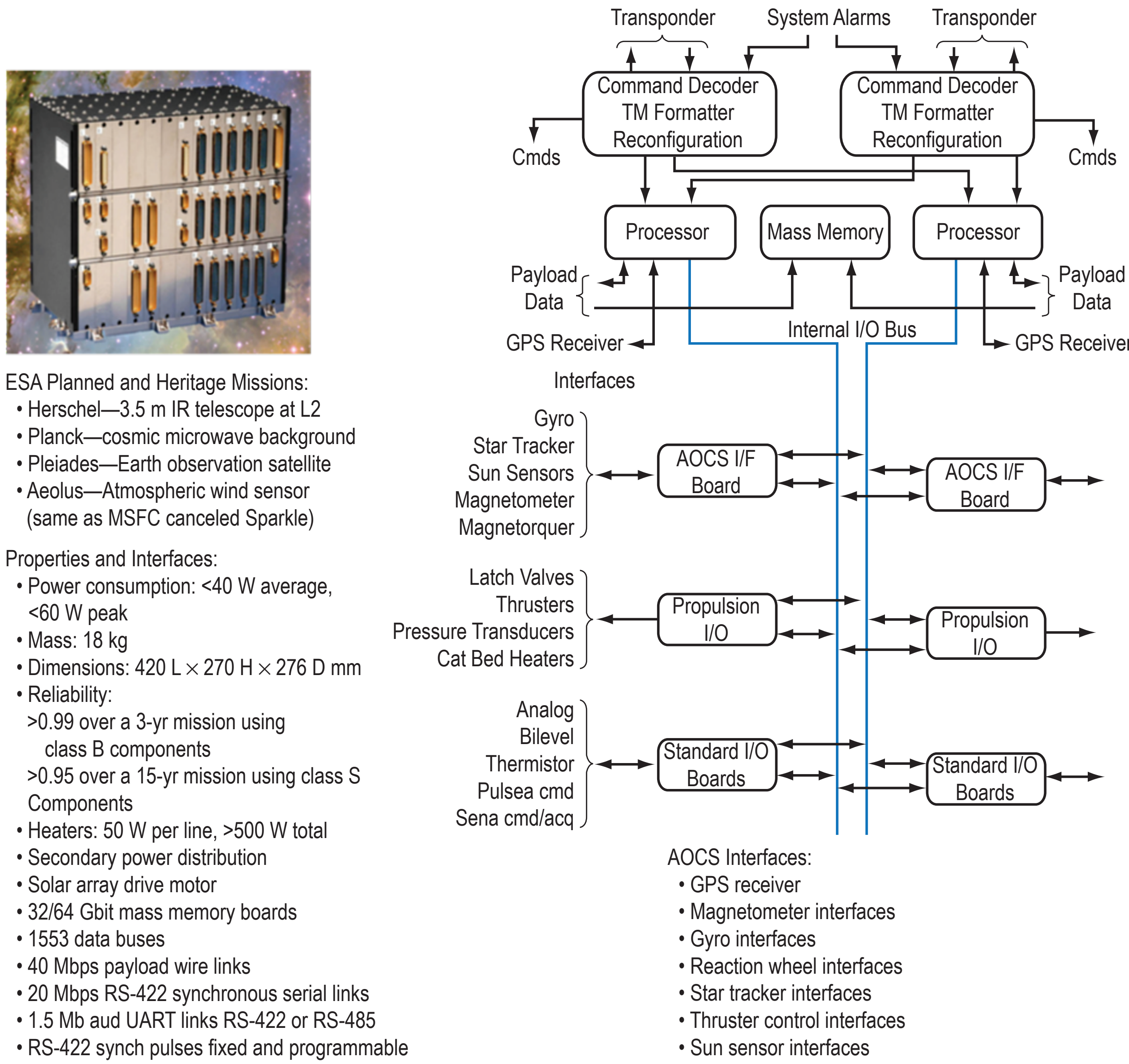


Figure 13. Saab Ericsson satellite computer.

The analyst assumed that the individual science instrument packages include all the required data processing, filtering, and buffering required, along with thermal, health, and status control. All science data are to be transmitted to the spacecraft computer via a dual redundant spacecraft data bus for storage and downloading to ground. Instrument health and status telemetry will be collected by a second dual redundant spacecraft data bus, processed, and stored independently of the science data. All science and telemetry data should be identified and time stamped for later correlation and downloading to ground.

A dual redundant primary power feed will be supplied to an instrument controller for each of the four major instruments. Those controllers must distribute secondary power to the instrument and instrument's electronic boxes, perform all required operations (e.g., safe mode), control any mechanism required (e.g., shutters), and perform the thermal management of the dedicated systems. All cabling between the controllers and science boxes were to be included in the science package mass estimates. A conceptual block diagram of the Xenia spacecraft showing the relative interfaces of the various avionics, communications, ACS, science instruments, and power subsystems is shown in figure 14. A summary of the subsystems, mass and power requirements is shown in table 12.

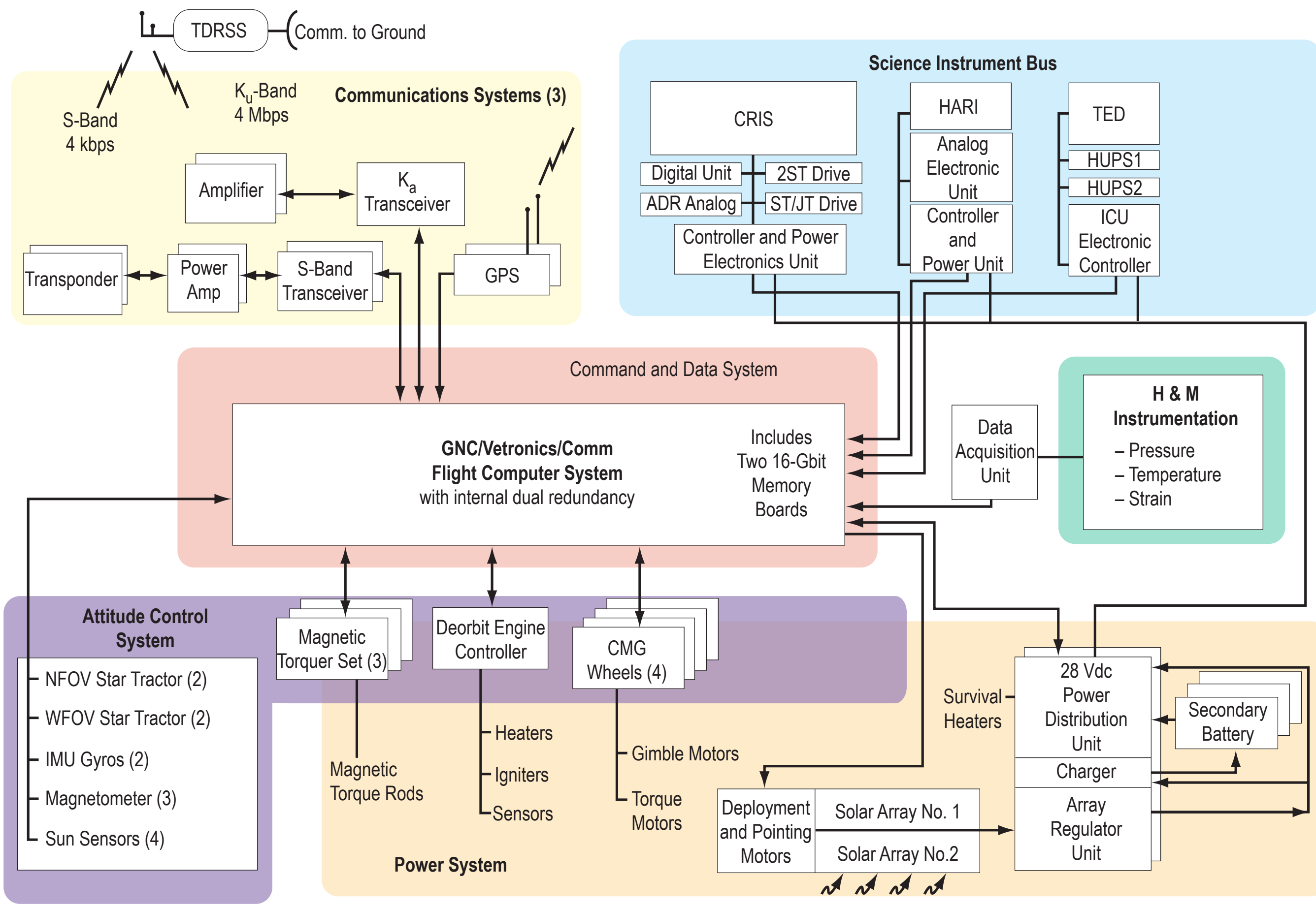


Figure 14. Conceptual block diagram of the Xenia spacecraft.

Table 12. Summary of subsystems' mass and power requirements.

Avionics	Mass (kg)	Power (W)
Attitude control system	320	240
Command and data system	22	107
Instrumentation and monitoring	5	7
Communications system	45	203
Avionics cabling	34	NA
Totals	426	557

7.2 Avionics Results and Conclusions

An omnidirectional K_u-underband communication link was chosen for simplicity and mass savings. A pointing antenna may be blocked by structures, restricting transmission capability. A link budget analysis was performed showing that a 36-Mbps omnilink could be made to the TDRSS with a 10-W transmitter.

A redundant 5-W, S-band system is used for command, telemetry, and GRB event notification links with the TDRSS. Fast, worldwide GRB notifications are made possible using the TDRSS MA mode. No direct link to ground is planned for normal operations; all links are through the TDRSS. The Saab Ericsson spacecraft computer has built-in redundancy, extra memory and speed capacity, and all the inputs and outputs required for this application, along with good heritage.

8. POWER

Power required for the Xenia spacecraft and its science package is detailed in table 13 (all requirements include 30% contingency).

Table 13. Power

	W
Attitude control system	239.9
Command and data handling	106.6
Communication power	202.8
Instrumentation power	6.5
Control power	83.2
Science instrumentation	1,388.0
Total	2,027

8.1 Ground Rules and Assumptions

The ground rules and assumptions used by the power subsystem are listed below:

- Mission
  - Orbit: 600-km circular
    - Period: 96.68 min
    - Max dark: 35.49
    - Min light: 61.19
  - Duration: 5 yr

- Environment
  - Solar power density: 1,370 W/m^2
  - Solar panel operating temperature: 76 °C
  - Thermal sink temperature: 279 K
  - Ambient electronics temperature : 30 °C max

- Secondary batteries
  - 25,000+ charge/discharge cycles
  - 40% max depth of discharge
  - Operating temperature: 30 °C max

- Solar arrays
  – 15% knockdown for cell mismatch, wiring, and self-shadow
  – 2.75%/yr degradation

- Charge/discharge
  – Charge voltage: >28.8 V
  – Discharge voltage: 28 V
  – Charge efficiency: 90%
  – Discharge efficiency: 90%

8.2 Design Highlights

Because of the relatively long science mission duration (5 yr) and the high levels of sunlight available, a solar-based power system was chosen for this spacecraft as shown in figure 15. This is a direct energy transfer design. The following are components as labeled in figure 15:

- Solar arrays
  – Type: GaAs three-junction cells on MJ55 Honeycomb substrate
  – Conversion efficiency (at op temp 76 °C): 24%
  – Specific power (end of life): 258.3 W/m^2
  – Areal density: 2.24 kg/m^2
  – Area: 14.65 m

- Array regulation unit (ARU)
  – Type: Sequential shunt regulator
  – Array interface: 60 strings
  – Switch modulation frequency: 50 kHz
  – Power delivery efficiency: 97.3%

- Charge/discharge unit
  – Type: Linear charge/discharge
  – End-to-end efficiency: 81%

- Battery
  – Type: Lithium (Li)-ion secondary batteries; based on Saft Li-ion VES-180 cells
  – Cell name-plate energy: 3.6 V, 50 Ahr at 30 °C operating temperature
  – Packing factor: Battery mass = 1.29 × combined cell mass
  – Series cells per battery: 8
  – Depth of discharge: 40% max
  – Battery unit effective capacity: 576 Whr
  – Battery unit mass: 11.66 kg
  – Number of units: 2

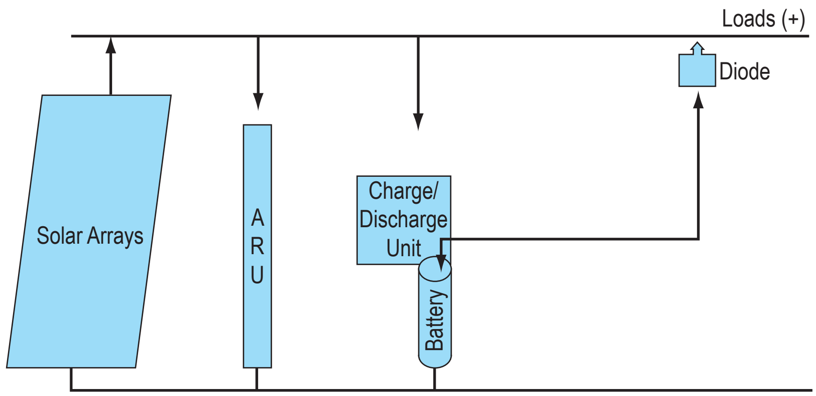


Figure 15. Solar-based power system.

Table 14 shows the power system masses.

Table 14. Power system masses.

Power Component	Mass (kg)
Power distribution unit	12.48
Cabling (5 m, redundant)	5.59
ARU	31.35
Solar array (14.65 m^2)	32.82
Secondary batteries (2)	23.32
Battery charger	63.97
Total power mass	169.53

9. THERMAL

A passive thermal design concept was developed for the Xenia spacecraft. Heat rejection of the subsystems and instrument power is accomplished by spacecraft radiators, heat pipes, silverized teflon tape, and closeout multilayer insulation. Spacecraft bus side and aft panels double as radiators to optimize mass as shown in figure 16. The bus outer surfaces are covered in low absorptivity silverized teflon in order to cold bias the spacecraft and minimize temperature fluctuations due to orbital position. To ensure a long life in the presence of atomic oxygen, the teflon will be coated with silicon oxide which acts as an atomic oxygen absorber.

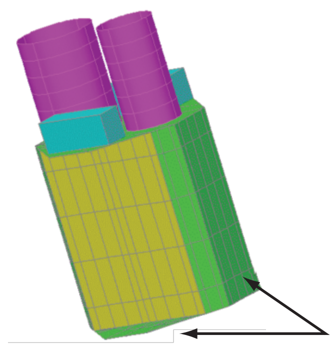


Figure 16. Space radiator panels.

A Thermal Desktop® model of the spacecraft structure was developed for the thermal analysis and design. The model was based on the FEMAP model of the spacecraft structure. A total of 2,024 W of spacecraft power/heat dissipation was considered in the thermal analysis, as shown in table 15. Heat loads were distributed on the spacecraft panels according to specific box locations, as shown in figure 17. Ammonia heat pipes with axially grooved tubing are to be placed in equipment panels to carry internal power loads to radiator surfaces, such as the 364 W attributed to the battery charger. Heat pipes are also to be used to isothermalize radiator panels. The aft octagonal bulkhead panel is also used as a radiator. A Sun avoidance angle of 45 deg will serve to ensure that at least half of the aft radiator panel provides an ideal sink to deep space. Side radiator panel performance will be degraded when exposed to solar heating for any extended period of time. The radiators have been oversized to account for degradation due to less than optimum deep-space viewing.

Steady-state analysis results indicate that spacecraft orbital average temperatures will remain between –10 and 34 °C for a maximum β angle of 33.5 deg as shown in figure 18. Orbital average temperatures of the spacecraft for a minimum β angle of zero degrees will be between

Table 15. Heat loads.

Component	Power or Heat Dissipation (W)
ACS/CDS	222
COMM	156
Power	585
CRIS	909
HARI	60
TED	92
Total	2,024

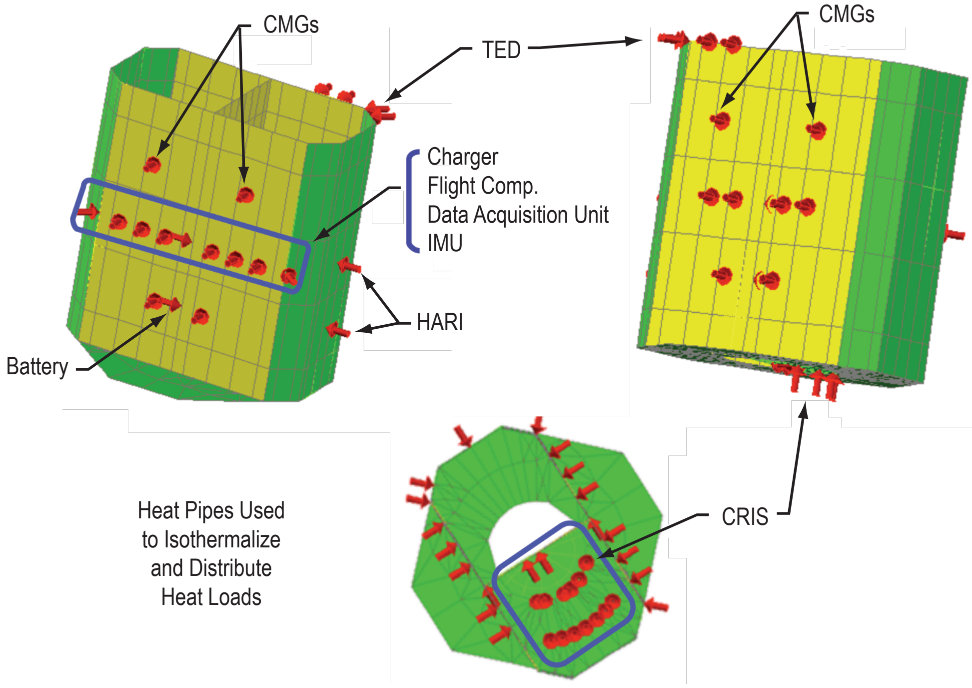


Figure 17. Heat loading for thermal analysis.

–9 and 35 °C. The side radiators were modeled in a worst-case position with one radiator facing the Sun, the other facing the Earth. The predicted temperatures are within nominal operating range for the spacecraft subsystems and instrument components.

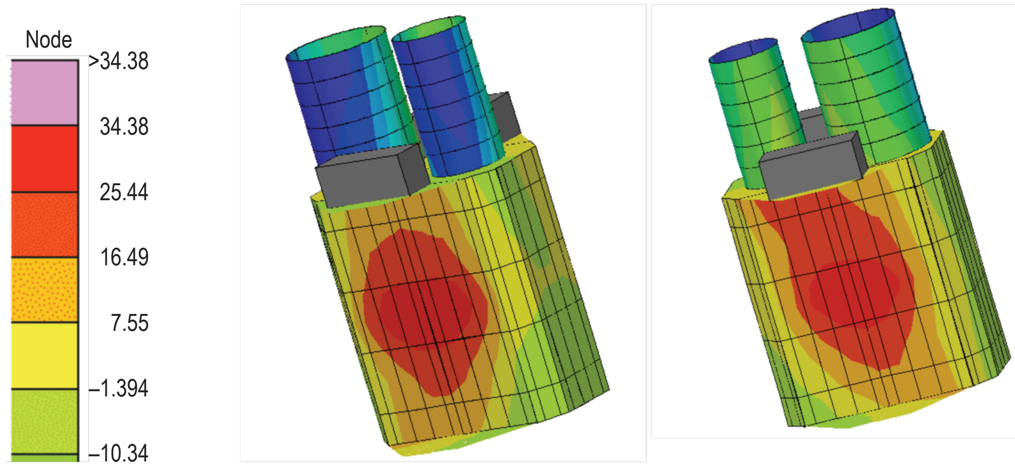


Figure 18. Spacecraft average temperatures, $\beta = 33.5$ deg.

Total thermal control mass is estimated at 32.7 kg as shown in table 16.

Table 16. Thermal control mass for the Xenia spacecraft.

Component	Units	Mass (kg)
Instrument light shield and baffle multilayer insulation	15 m^2	5.0
Closeout blankets	15 m^2	7.5
Heat pipes	4 @ 1.3 kg each	5.2
Silverized teflon tape	25 @ 0.6 kg/m^2	15.0
Total		32.7

10. PROPULSION

The chosen deorbit propulsion system for the Xenia spacecraft consists of one Aerojet R42 bipropellant spacecraft engine, one propellant tank for each propellant (nitrogen tetroxide oxidizer (NTO)/MMH), and a separate helium pressurization system for each propellant tank. Several trade studies were performed during this conceptual design study to determine the appropriate deorbit propulsion system for the Xenia spacecraft. The total calculated wet mass for the propulsion system was 166.4 kg.

10.1 Assumptions

Mission analysis was performed to determine the required ΔV the spacecraft needed to achieve to provide for a proper deorbit sequence once the scientific portion of the Xenia spacecraft mission was complete. From the data provided, the ΔV required was 163 m/s with a T/W of >0.025. The initial mass of the spacecraft continued to be in flux due to changes to other spacecraft subsystems, therefore analysis was performed at several different masses in order to determine the appropriate propulsion system to use. For each propulsion system analyzed, data were obtained from the spacecraft engine vendor (either aerojet or ATK) and used in the analysis. It was assumed that the propulsion system would be required to be stored for at least 5 yr on orbit and would be in an orbit that required no external thermal protection (no heaters required).

10.2 Solid Versus Liquid Trade Study

The first trade study performed was to determine the effect of a solid propulsion system versus a liquid propulsion system. For this study, two liquid systems were used—hydrazine (N_2H_4) monopropellant and NTO/N_2H_4 bipropellant. The solid motor propulsion system was chosen from data obtained from the ATK space products catalog. Using the assumptions above and assuming that the propellant tank pressure was 125 psia above the quoted spacecraft engine pressure and that the pressure in the helium tanks was 4,500 psia, data were obtained for initial spacecraft masses ranging from 1,500 to 2,000 kg as shown in figure 19.

As shown in figure 19, the N_2H_4 monopropellant system has a higher wet mass than the solid or the NTO/N_2H_4 bipropellant system. The solid propulsion system tends to be higher than the NTO/N_2H_4 system and the NTO/N_2H_4 system tends to be the lightest propulsion system. However, the difference between the solid propulsion system and the NTO/N_2H_4 propulsion system is not high. This shows that the high propellant mass fraction of the solid can perform well against a higher performance (higher I_{sp}) bipropellant system in some cases. Notice that as the initial spacecraft mass increases, the difference between the solid and liquid system also increases, showing that with higher energy (ΔV and M_0) systems, the higher performance system is the least massive. With this result and the assumption of a 5-yr storage requirement, a bipropellant engine system was chosen as the baseline deorbit propulsion system for the Xenia spacecraft.

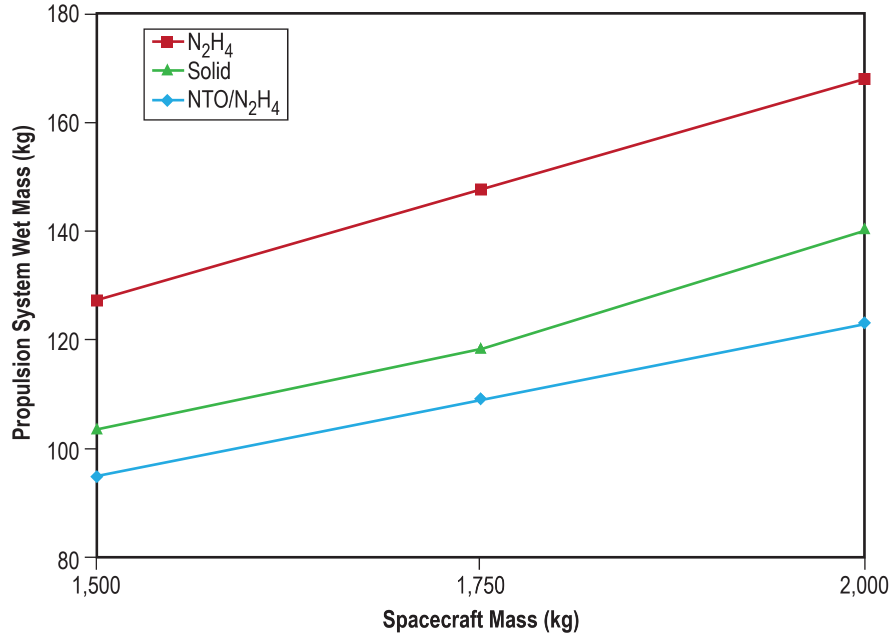


Figure 19. Solid versus liquid trade study results.

10.3 Liquid Engine Trade Study

Once the propulsion system type had been chosen, data were obtained on several candidate bipropellant propulsion systems to be traded. Table 17 describes the candidate engines and their characteristics used in this analysis.

Table 17. Bipropellant engine characteristics.

Engine	HiPat	R4D	R4D	R42
Propellant	NTO/N_2H_4	NTO/MMH	NTO/MMH	NTO/MMH
Thrust (lbf)	100	110	110	200
Chamber pressure (psia)	137	108	108	103
Mixture ratio	0.86	1.65	1.65	1.65
Expansion ratio	300	44	164	160
Specific impulse (s)	326	300	311	303
Mass (kg)	5.2	3.4	3.76	4.53

Using the assumptions above and negating the effect of the propellant tank due to the fact that the structures and configuration disciplines need to create cylindrical tanks instead of spherical due to volumetric constraints, it was found that although the NTO/MMH propulsion systems have a lower performance than the HiPat NTO/ N_2H_4 system, the higher engine mass and required helium negates this effect as shown in figure 20. Therefore, the R42 engine was chosen as the liquid propulsion system engine for the Xenia spacecraft deorbit propulsion system.

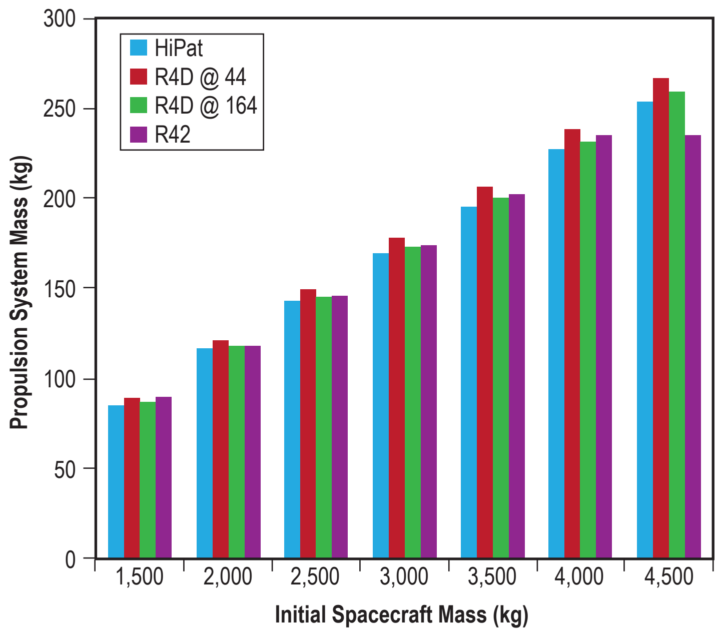


Figure 20. Liquid engine trade study results.

10.4 Propulsion Conclusion

With each of these propulsion system trade studies complete, the Xenia deorbit propulsion system was chosen. The system chosen was an NTO/MMH R42 propulsion system that included separate cylindrical propellant tanks as well as separate pressurization systems. Table 18 outlines the total mass of the deorbit system chosen.

Table 18. Xenia deorbit propulsion system mass statement.

Work Breakdown Structure Element—Descent Stage				Quantity	Unit Mass (kg)	Total Mass (kg)
2.0	Propulsion					15.50
	2.1	Main engines		1	5.00	5.00
	2.2	Fuel tank		1	3.30	3.30
	2.3	Main oxidizer tank		1	3.60	3.60
	2.4	Pressurization tank		2	1.30	2.60
	2.5	Feed system		1	1.00	1.00
8.0	Growth					3.10
	8.3	Propulsion		20%		3.10
Dry mass						18.60
9.0	Noncargo					0.47
	9.1	Propellant residuals		1		0.00
		9.1.1	Fuel	1	2.78	2.78
		9.1.2	Oxidizer	1	4.59	4.59
	9.2	Pressurant				0.47
		9.2.1	Fuel	1	0.23	0.23
		9.2.2	Oxidizer	1	0.24	0.24
Inert mass						0.47
Total less propellant						19.07
12.0	Propellant					147.33
	12.1	Main fuel		1	55.60	55.60
	12.2	Main oxidizer		1	91.73	91.73
Gross mass						166.40

11. STRUCTURES

Since the Falcon 9 launch platform is large enough for the Xenia spacecraft, the structural requirements of the spacecraft bus are driven by the primary science instruments' shape, size, and mass. The two large telescopes, the HARI and CRIS, are symmetrically arranged about the bus fore-aft axis, but the mass distribution along their respective axes varies in vertical location. The two telescope mirror assemblies and the CRIS cryogenic dewar require unique secondary structural support. The spacecraft concept uses lightweight aluminum panels, tubing struts, and I-beam structural supports. 2024–T351 plate was used for the radiator panels and 7075–T651 bar was chosen for struts and the supporting I-beam cage to optimize thermal properties and strength, respectively.

A rectangular bus structure provides the needed real estate and orientation to utilize interior and exterior panels for component mounting that incorporate heat pipes to act as radiators for thermal management. The aft octagonal bulkhead panels also will be used for thermal management. To incorporate heat pipe channels, a thickness of ⅛ to ¼ in was modeled, resulting in very high margins of safety (10 to100). This thickness was estimated to approximate the final mass and structural strength after heat channel machining.

A closeout truss structure of 0.060 gauge (0.0015 m) round tubing struts encloses the intermediate bulkheads which gives the bus structure an octagonal shape. Thermal insulation will close out these largely nonstructural volumes. Midway fore and aft on two opposite sides of the bus, telescoping booms support the foldout solar arrays and vibration dampers minimize oscillations after fast slew. These were sized similarly to those used on the Hubble telescope, but proportionally less massive.[4]

A Finite Element Modeling and Postprocessing (FEMAP) model was created based on the Pro-E configuration and loaded with components and instrument masses and analyzed with NX Nastran. Eight load cases were run with 5-g axial and 0.9-g lateral loads at 45-deg intervals. A 1.4 factor of safety for isotopic strength and a 0.65-buckling factor were used in the analysis. Once the structural analysis was successful, the model was optimized using Hypersizer® and HyperFEA® for minimum mass. Structural members were grouped, results were analyzed, and material thicknesses were adjusted to reflect common raw material stock and other manufacturability criteria. The results were iterated through FEMAP until a minimum mass and positive margin of safety was converged upon using a compressive yield stress of 2.69×10^{-8} Pa for aluminum.

Table 19 summarizes the structural mass of the spacecraft bus and secondary structure to support propulsion, power components, and science instruments. The FEMAP Finite Element Analysis (FEA) model used to analyze and size the spacecraft is depicted in figure 21. Plates, bars, and rigid mass elements were used to represent all mass estimated by all the disciplines developing the spacecraft.

Table 19. Structural mass of the Xenia spacecraft.

Structural Mass (All Aluminum)		Quantity	Total	399 kg
Secondary	Solar array dampers, actuators, and booms	2	30	60
Secondary	Propulsion			15
Secondary	Science instruments			99
Primary	Spacecraft bus			225

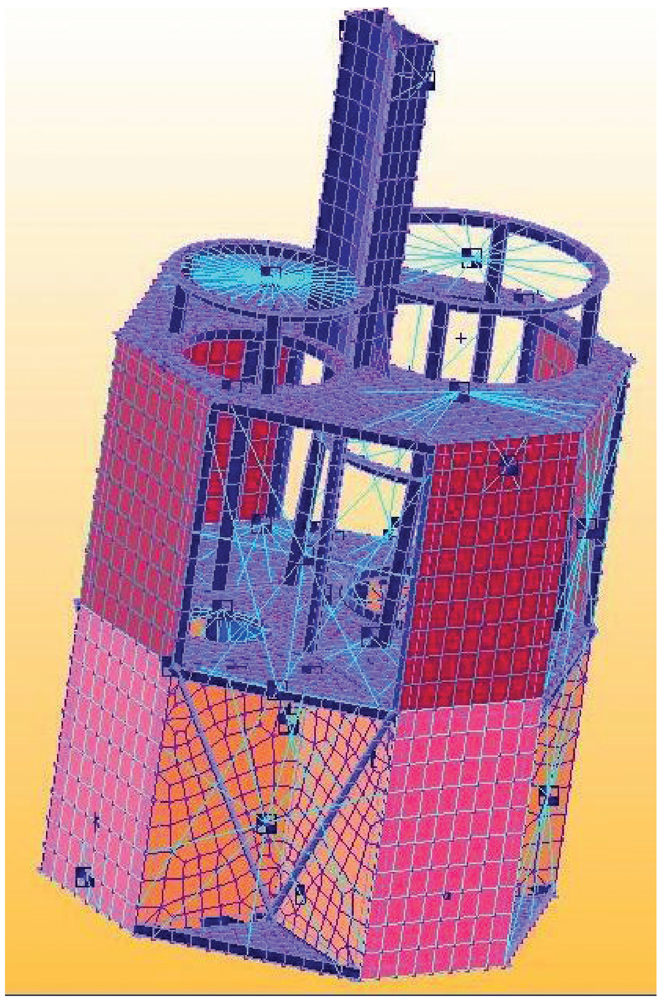


Figure 21. FEMAP FEA model of Xenia spacecraft.

12. CONCLUSION

The above summarizes the work of an MSFC Advanced Concepts Office (ACO) study, the purpose of which was to complete a conceptual spacecraft design for the Xenia mission. Given the science requirements and conceptual designs for the science instruments, ACO completed the mission analysis and created a conceptual spacecraft design that will meet the science goals. The resulting spacecraft meets all launch, mission, thermal, communication, end-of-life, pointing, slewing, structural, and power requirements. Of primary concern were the thermal and pointing/slewing requirements. The compact spacecraft volume coupled with the large thermal loads from the science instruments and spacecraft subsystems was an area of concern for thermal control, but ACO analysis shows that the thermal requirements are met with the proposed spacecraft design. In addition, while previous spacecraft concepts that used reaction wheels could not meet the rapid slewing and repointing requirement for Xenia, the current design incorporates control moment gyros. This approach uses less power for slewing and can also rotate the spacecraft much more quickly, allowing the spacecraft to meet the rapid slewing requirement.

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14. ABSTRACT
The proposed Xenia mission will, for the first time, chart the chemical and dynamical state of the majority of baryonic matter in the universe. Using high-resolution spectroscopy, Xenia will collect essential information from major traces of the formation and evolution of structures from the early universe to the present time. The mission is based on innovative instrumental and observational approaches: observing with fast reaction gamma-ray bursts (GRBs) with a high spectral resolution. This enables the study of their (star-forming) environment from the dark to the local universe and the use of GRBs as backlight of large-scale cosmological structures, observing and surveying extended sources with high sensitivity using two wide field-of-view x-ray telescopes—one with a high angular resolution and the other with a high spectral resolution.

This Technical Memorandum describes the Xenia spacecraft concept at prephase A level.

15. SUBJECT TERMS
astrophysics mission concepts, Xenia, spacecraft thermal control, spacecraft, control moment gyro, baryonic matter, gamma-ray bursts, large-scale cosmological structure, rapid slewing, space telescope

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